Shock-induced separated flows on the lee surface of delta wings

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SUMMARY

Experiments were conducted to study shock-induced separated flows on the lee surface of delta wings with sharp leading edge at supersonic speeds. Two sets of delta wings of different thickness (10° and 25° normal angle), each with leading edge sweep angles varying from 45° to 70°, were tested. The measurements, carried out in a Mach number range from 1·4 to 3·0, included oil flow visualisations (on both sets of wings) and static pressure distributions (on the thicker wings only). Using the test results, some features of shock-induced separated flows, including in particular the boundary between this type of flow and fully attached flow, have been determined. The experimental results indicate that this boundary does not seem to show any significant dependence on wing thickness within the limit of thicknesses tested. It is shown that this boundary can be predicted for thin delta wings using a well known criterion for incipient separation in a glancing shock wave boundary layer interaction, namely that a pressure rise of 1·5 is required across the shock. Comparison of the predicted boundary with experimental results (from oil flow visualisations) shows good agreement.

NOTATION

b	Wing local semispan
C_p	Pressure coefficient ($= (p - p_\infty)/1/2\gamma p_\infty M_\infty^2$)
K	Constant defined by equation (5)
M	Mach number
p	Static pressure
y	Spanwise distance measured from wing centre line
α	Angle of attack
γ	Ratio of specific heats
δ	Wedge angle
υ	Prandtl-Meyer angle
θ	Flow deflection angle (Fig. 10)
ϕ	Yaw angle
Λ	Sweepback angle of leading edge
η	y/b

Subscripts

∞	Freestream condition
N	Quantity measured normal to leading edge
T	Quantity measured along the leading edge
1,2	Upstream and downstream of swept P-M fan, respectively
3	Downstream of embedded shock

1. INTRODUCTION

The flow on the lee side of delta wings has been the subject of considerable study over the past three decades. These studies have shown that depending on the freestream and wing parameters, a fascinating variety of different flow types is possible on the lee surface. Broadly, such flows can be classified into two categories — those which are attached at the leading edge and those which are separated. After examining all the data available up to 1962, Stanbrook and Squire[1] defined a boundary between these two types of flow. This 'Stanbrook-Squire' boundary, shown in Fig. 1 for wings with sharp leading edge, is defined in a plane of Mach number (M_N) and angle of attack (α_N) normal to the leading edge. The fact that this boundary is not sharp but a band has been attributed variously to Reynolds number and wing thickness effects, and to difficulty in identifying the type of flow from experiments[2].

Detailed experiments were carried out by Szodruch[3] and Miller and Wood[4] to study the types of flow that occur on the lee surface of thick and thin delta wings respectively. Both these studies have identified several additional types of flow containing various combinations of vortices and shock waves. A comparison of the flow types that occur and the boundaries between them reported in these two investigations (Figs. 2(a), 2(b)*) indicates certain basic differences. One such important difference is that while Miller and Wood[4] clearly identify on a thin wing a flow type without separation, Szodruch[3] on a thick wing does not. Also, in the study of Miller and Wood, shock-induced separation occurs for $\alpha_N >$ 14° and for all normal Mach numbers to the right of the Stanbrook-Squire boundary. On the other hand, Szodruch indentifies shock induced separated flow only for normal Mach numbers greater than approximately 1·7 and normal angles of attack significantly lower than 15°. It is not clear if all these differences can be attributed to wing thickness effects, particularly since the difference in the wing thickness between the two studies is not large (normal angles, δ_N, of 10° in Miller and Wood's tests compared with 26° in Szodruch's

*Figures 2(a) and 2(b) display only some of the flow types observed by Szodruch, Miller and Wood and which are relevant to the present paper.

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tests). It is probable that these differences may in part be due to difficulties in interpretation of experimental results, most of which are flow visualisation data (oil flow, vapour screen, etc). The present authors are investigating this aspect through a systematic set of measurements covering a fairly wide range of Mach number, sweep back of the leading edge and angle of attack.

Notwithstanding these differences, the Stanbrook-Squire boundary seems to be fairly well established for thin wings from the original work of Stanbrook and Squire[1] and the measurements of Miller and Wood[4] (and confirmed by the present study). Miller and Wood have obtained a boundary between fully attached and shock-induced separated flows for thin wings. One of the aims of the present study is to confirm the location of this boundary and more importantly study the effect of wing thickness. Squire[5] has made an attempt to predict the Stanbrook-Squire boundary based on the assumption that the type of flow on the lee surface of a delta wing at supersonic speeds is determined by the nature of the flow on the windward surface, in particular, leading edge separation occurring when the attachment lines move inboard from close to the leading edge towards the centre line of the wing (on its windward surface). There is some experimental evidence[5] to support this view. Squire used thin shock layer theory to predict the conditions under which the attachment lines move. The boundary so calculated is in reasonable agreement with the Stanbrook-Squire boundary over a range of α_N which is dependent on the wing thickness. The agreement is good up to $\alpha_N \simeq 40°$ and 15° for thin ($\delta_N = 10°$) and thick wings ($\delta_N = 30°$) respectively. More recently, Squire[2] carried out experiments on a delta wing with elliptic cross section and concluded that leading edge separation might occur if the calculated pressure jump across the upper surface cross flow shock is more than about 2. Both these predictions need to be validated for a wider range of wing and free stream parameters.

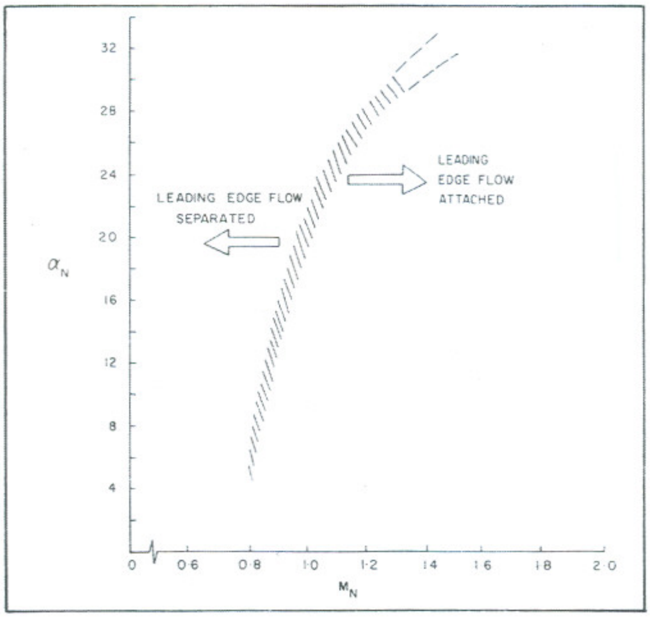


Figure 1. Stanbrook-Squire boundary for thin wings.

While some attempts have thus been made to predict the Stanbrook-Squire boundary, no systematic work seems to have been carried out in trying to predict the boundary between attached flows and flows with inboard shock induced separation. This paper describes a set of experiments carried out to study some features of shock induced separated flow, and in particular to determine the effect of wing thickness on the boundary between attached and shock induced separated flow.

A simple method, based on the proposal that there is an analogy between shock wave boundary layer interaction on a delta wing lee side and glancing shock wave boundary layer interaction on a flat plate is described to predict this boundary.

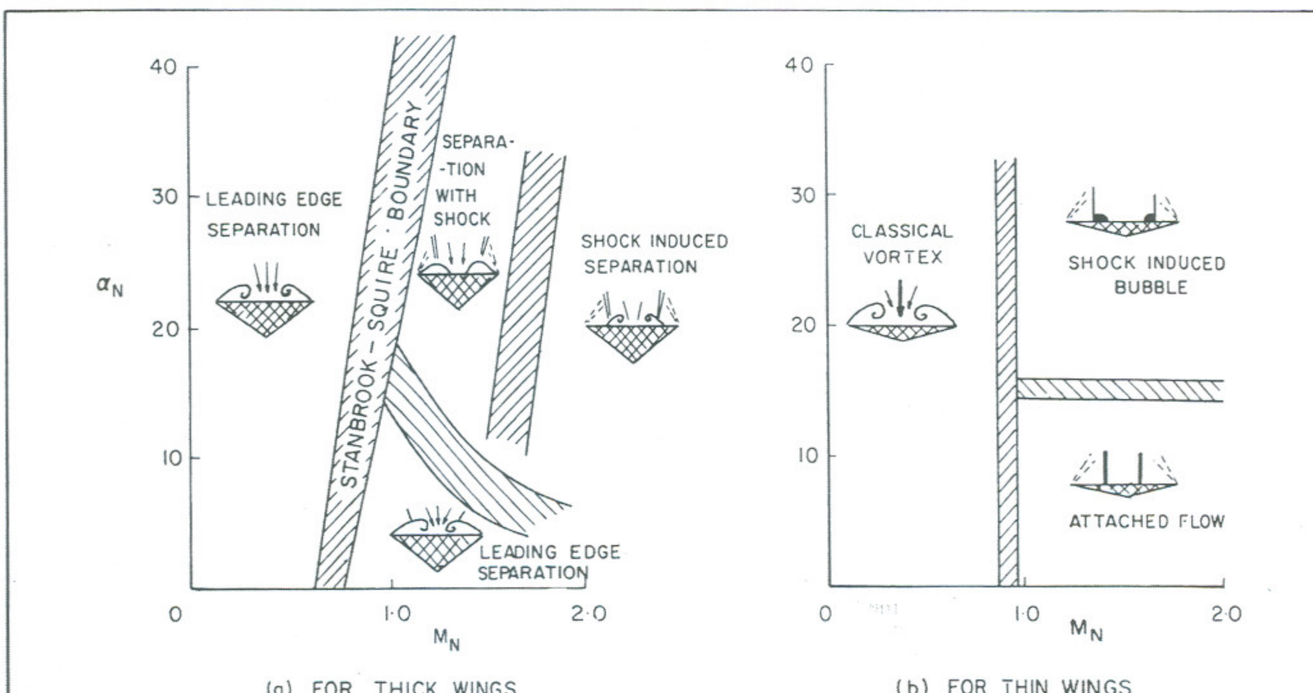


Figure 2. Some types of flow on thick[3] and thin[4] wings.

2. EXPERIMENTAL FACILITY AND MODELS

The experiments were conducted in the 0·3 M trisonic wind tunnel of the National Aeronautical Laboratory. This facility is of blowdown type using compressed air at ambient temperature of 300°K and a maximum reservoir pressure of about 11 atmospheres. The facility has a Mach number range of 0·5 to 3·0.

Two types of model support system were used depending on the Mach number and angle of attack. A conventional centre mounted support system was used up to Mach number of 2 and angles of attack up to 15°. For higher Mach numbers and/or higher angles of attack, a side wall mounted model injection-retraction system was used.

The delta wings tested had leading edge sweep angles of 45°, 50°, 60 and 70° (Fig. 3). All the wings had a triangular cross section with flat upper surface and sharp leading edges. For each sweep angle, two wings were made; one had a wedge angle normal to the leading edge of 10° while the other had an angle of 25°. According to a classification given by Szodruch and Peake[6], the 10° normal angle wing is a thin wing while the 25° one is a thick wing.

The leeward surface of the wings having $\delta_N = 25°$ was instrumented with a spanwise row of several pressure orifices located well upstream of the trailing edge. Figure 3 shows the details of the orifice location.

Table 1 shows the range of test parameters covered in this study. It may be noted that, while the models having $\delta_N = 25°$ were used for both pressure measurements and oil flow visualisation, the models having $\delta_N = 10°$ were used for oil flow visualisation only. Oil flow visualisations were carried out using a mixture of titanium dioxide in vacuum pump oil and oleic acid. Pressure data were obtained from a 48 port scanivalve-transducer system mounted outside the tunnel.

3. EXPERIMENTAL RESULTS AND DISCUSSION

3.1 Boundary between attached type flows and flows with inboard shock-induced separation

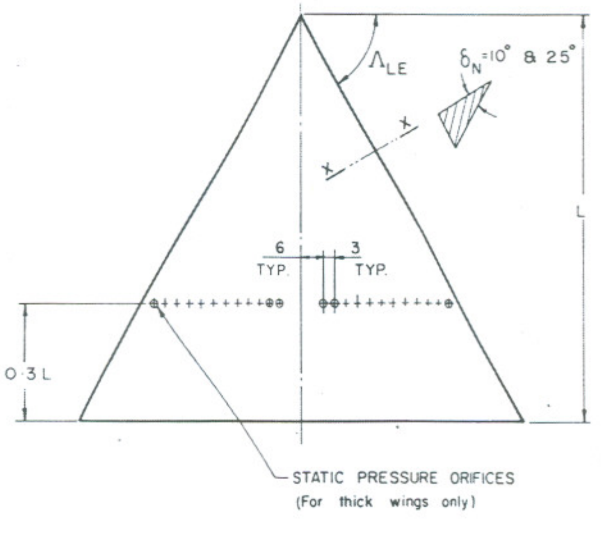
The type of flow which occurred on the model was identified essentially by the surface oil flow pattern. Separation was identified using the criterion given by Maskell[7] that a separation line is an envelope of converging surface streamlines. Figure 4 shows four sets of oil flow patterns obtained on 60°, 50° and 45° sweepback thin wings, covering a M_N range of about 1·1 to 2·1. The angles of attack for each case have been so chosen that their lower and higher values are the highest and lowest respectively at which attached and inboard separated flow are definitely observed. Figure 4(a) shows the oil flow patterns on the 60° sweepback thin wing at a Mach number of 2·08 and at angles of attack of 4° ($M_N = 1·06$; $\alpha_N = 8°$) and 6° ($M_N = 1·06$; $\alpha_N = 12°$). Figure 4(b) shows the oil flow patterns on the same wing at a Mach number of 2·94 and at angles of attack of 6° ($M_N = 1·5$; $\alpha_N = 12°$) and 8° ($M_N = 1·5$; $\alpha_N = 15·7°$). Figure 4(c) shows the oil flow patterns on a 50° sweepback wing at a Mach number of 2·46 and at angles of attack of 8° ($M_N = 1·61$; $\alpha_N = 12·3°$) and 10° ($M_N = 1·61$; $\alpha_N = 15·3°$). Figure 4(d) shows the oil flow patterns on a 45° sweepback wing at a Mach number of 2·94 and at angles of attack of 10° ($M_N = 2·1$; $\alpha_N = 14°$) and 12° ($M_N = 2·13$; $\alpha_N = 16·7°$). It can be seen from these oil flow patterns that the flow at the lower angles of attack in each case corresponds to fully attached flow whereas the flow at the higher angles of attack corresponds to the inboard separated case.

Results obtained from these and other oil flow patterns on thin wings are plotted in Fig. 5 where the open and closed symbols correspond to attached and inboard separated flows respectively. The shaded band shown in the figure is the boundary which separates these two types of flow. The boundary is drawn such that no case of attached flow exists above it and also no case of separated flow exists below it. Within the shaded band, both types of flow exist or the oil flow pattern is not clear enough to identify the type of flow conclusively. Results of some tests made with the wing yawed to the freestream are also shown in the above figure where an effective sweep ($\Lambda \pm \phi$, where ϕ is the yaw angle) is used in estimating M_N and α_N. Also plotted in Fig. 5 are the results of Miller and Wood[4] and Rein[8]. While Rein's tests were on a 52·5° sweep wing with a normal angle of 11°, Miller and Wood's tests were on a family of wings of 10° normal angle

TABLE 1

M_∞	Blowing Pressure in atm	Reynolds Number per metre	Reynolds Number based on chord (millions)
1·6	2·35	30·38	4·63
1·8	2·84	34·45	5·25
2·0	3·05	34·11	5·20
2·2	3·41	34·88	5·32
2·5	5·15	45·53	6·94
3·0	6·74	46·12	7·03

Note: α Range for all tests 0 to 14°



MODEL DETAILS

Λ_{LE}	45	50	60	70
L	86·6	103·2	150	150

NOTE : All dimensions are in mm.

Figure 3. Details of delta wing models.

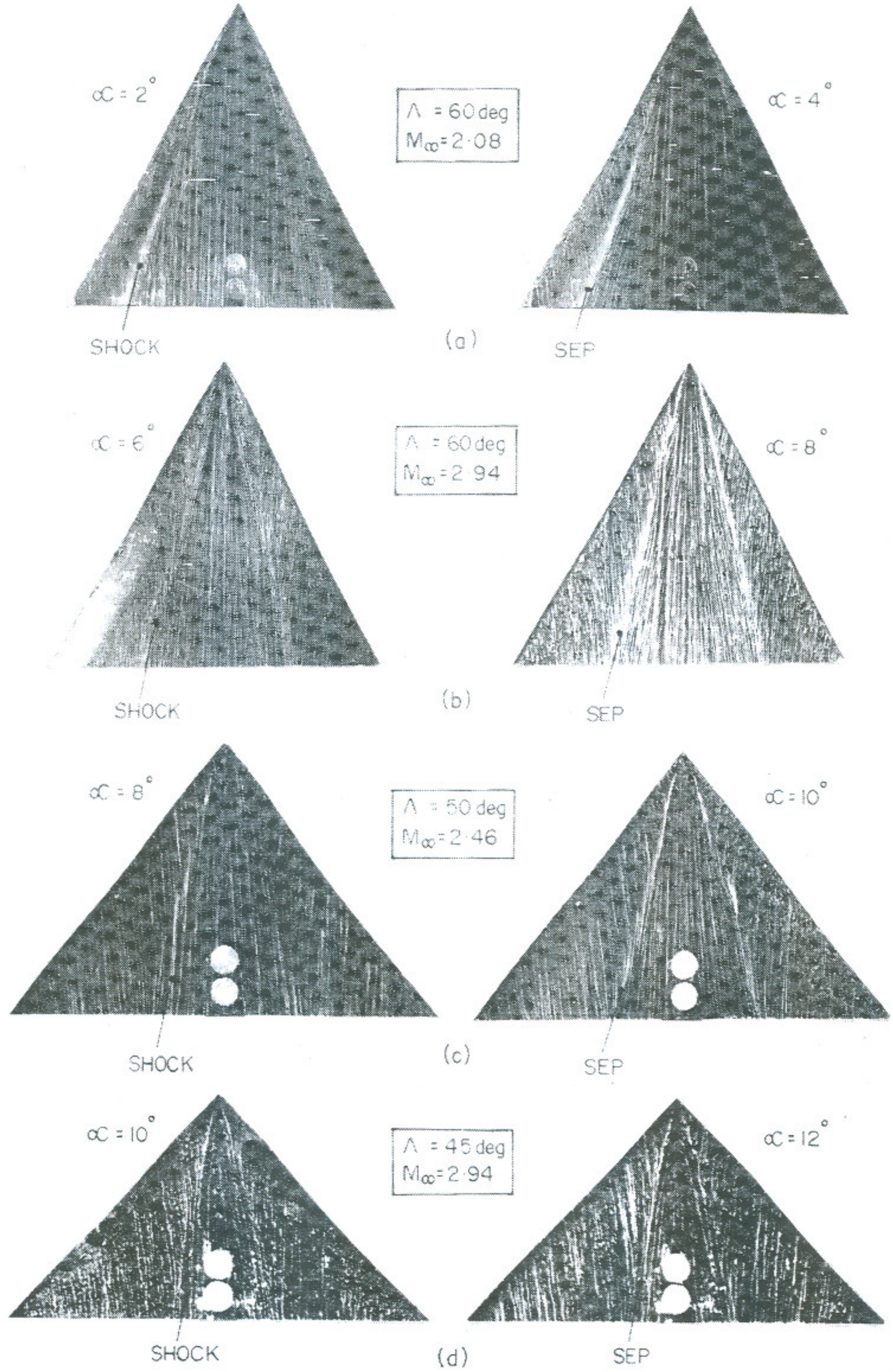


Figure 4. Typical oil flow patterns on thin wings.

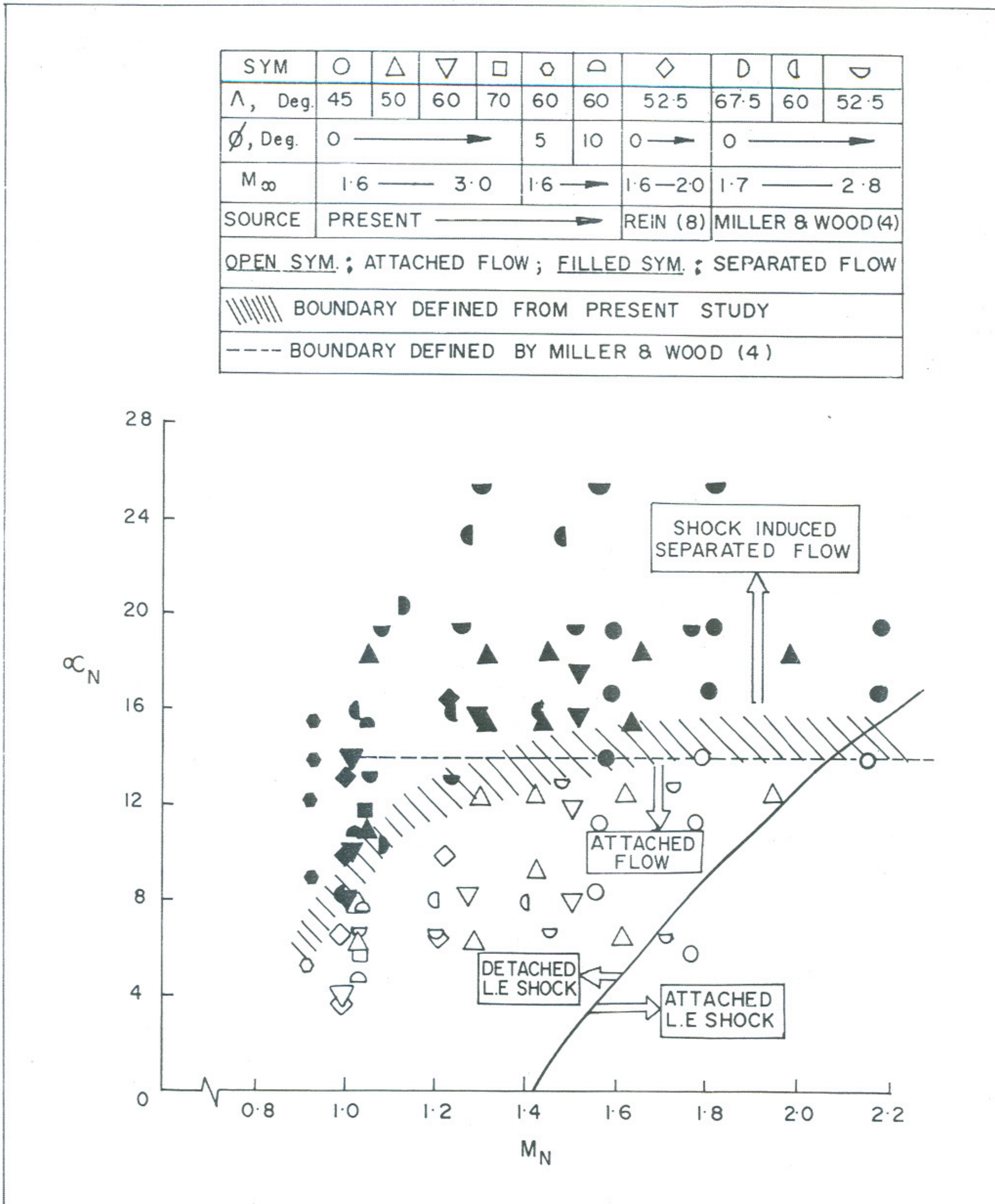


Figure 5. Experimental results from oil flow tests for thin wings ($\delta_N = 10°$).

with sweep angles from 52·5 to 75°. It is seen that all these results are also consistent with the boundary defined in the present study. Interestingly, Fig. 5 also shows that the boundary approaches a mean value of α_N of approximately 15° for $M_N > 1{\cdot}6$. The boundary as defined by Miller and Wood on the other hand corresponds to $\alpha_N = 14°$ for $M_N > 1$.

Figure 6 shows experimental results of surface oil flow pattern for a thick wing ($\delta_N = 25°$). The boundary defined in Fig. 5 for thin wings is plotted in this figure for comparison. It is seen that this boundary (defined for thin wings) is also valid for thick wings, indicating thereby that the boundary between these two types of flow is insensitive to wing thickness, at least within the range of thicknesses tested (ie, $\delta_N = 10°, 25°$).

The boundary between attached flow and inboard separated flow has been defined as a band with a width of about 2° in α_N. The technique (oil flow visualisation) of determining the type of flow precludes any more precise definition of the boundary; the width of the boundary in fact denotes the uncertainty in interpreting the oil flow patterns at conditions close to change of flow from one type to the other.

3.2 Separation line location

The spanwise location of the separation line was measured from oil flow visualisation photographs. Figure 7 shows the variation of separation line with angle of attack for wings of 50° and 60° sweep and at various Mach numbers. Data are plotted for wings of $\delta_N = 10°$ and 25° to see the effect of wing thickness. In general, it is seen that the separation line does not move outboard significantly as the angle of attack is increased; the change from an inboard separated flow to a classical leading edge separated flow* occurs over a very narrow α-range. The precise manner in which this change over takes place is not however known at the present time. The variation of the separation line position with Mach number and sweepback angle is as expected; an increase in Mach number or a decrease in sweep angle causes separation to move inboard.

Increase of wing thickness from $\delta_N = 10°$ to $\delta_N = 25°$ does not seem to cause an appreciable change in the mean position of separation; however change over from inboard separation to leading edge separation occurs at a smaller α for the thicker wing.

3.3 Static pressure distribution

Figure 8 shows three sets of static pressure distributions plotted in terms of the pressure coefficient C_p and spanwise position $\eta(= y/b$ where y is the distance from the centreline and b is the semi-span). The pressure distributions are on a 50° sweep wing at freestream Mach numbers of 2·46, 2·08 and 1·8.

Figure 8(a) shows the pressure distribution at a Mach number of 2·46 and is typical of flows with $M_N > 1{\cdot}6$. The distribution shows a fairly extensive uniform pressure region inboard of the leading edge, a pressure rise through the inboard embedded shock wave followed by another uniform pressure region. For $\alpha > 9°$, the pressure distribution shows a 'plateau' region or a region of reduced pressure gradient characteristic of flows with separation. The position of the separation line as measured from surface oil flow patterns is also shown in the figure by appropriately labelled arrows.

Figures 8(b) and 8(c) show pressure distributions at Mach numbers of 2·08 and 1·8 respectively. These are typical of flows with normal Mach numbers below about 1·6. These pressure distributions are characterised by a region of increasing pressure near the leading edge followed by a narrow extent of uniform pressure region just upstream of the embedded shock. The plateau (or region of diminished pressure gradient) indicating separation is evident for certain cases of M_∞ and α, but certainly much less clear here than for normal Mach numbers greater than 1·6 (Fig. 8(a)). The increasing pressure downstream of the leading edge could be due to the reattachment of the flow following the leading edge bubble.

4. PREDICTION OF INCIPIENT SEPARATION ON THE DELTA WING LEE SURFACE

The flow on the lee surface of delta wings for normal Mach numbers to the right of the Stanbrook-Squire boundary is characterised by the presence of an 'embedded' shock wave, the interaction of which with the boundary layer might result in separation if the normal angle of attack is sufficiently large (ie, above the shaded band of Fig. 5).

We propose that this interaction between the embedded shock wave and the leeside boundary layer is similar to a glancing interaction between an oblique shock wave and boundary layer on a flat plate. The possibility that these two flows might be similar was first suggested by Dunavant *et al*[9]. (Rough estimates of the boundary between attached flow and flow with shock induced separation on a delta wing were also made using this proposal; however a systematic study does not seem to have been made.) The similarity between the two flows is indicated in Fig. 9 which shows sketches of the surface flow patterns of the two flow fields. Also shown in the figure are the corresponding cross flows inferred from the surface flow patterns. The flow pattern in the glancing shock interaction is taken from Freeman and Korkegi[10] while the delta wing leeside flow pattern is from the present measurements. The figure shows that although the mechanism of generation of the shock wave is different in the two cases, there is a large measure of similarity in the two flow fields. Based on this, quantitatively also, one may expect certain overall features like, in particular, conditions for incipient separation to be the same for the two cases.

Korkegi[11] after examining a large amount of data, has given a correlation for incipient separation of a turbulent boundary layer in a glancing interaction. This correlation simply states that incipient separation occurs if the pressure rise across the shock wave is 1·5. This criterion for incipient separation in a glancing interaction may also be expected to be valid for the incipient separation of the boundary layer on the delta wing leeside. The conditions under which the leeside boundary layer might separate can then be determined as follows.

Figure 10 shows details of the flow normal to the leading edge on the lee surface of a delta wing. It is assumed here that the windward side shock wave is attached to the leading edge. Analysis of the case where the leading edge shock is detached poses some problems essentially because the shock shape and thus the flow downstream of the shock cannot be calculated easily. However, as will be shown subsequently, the predicted boundary between attached and inboard separated flows does not seem to be dependent, at least to a first order, on whether the leading edge shock is attached or detached.

*A distinction is made between flows characterised by a stable leading edge vortex and with leading edge separation bubble. Flows which have a leading edge bubble are considered here as attached flows since the oil flow photographs indicate a large region of attached flow following the bubble. Flows with stable leading edge vortices are termed as classical leading edge separated flows.

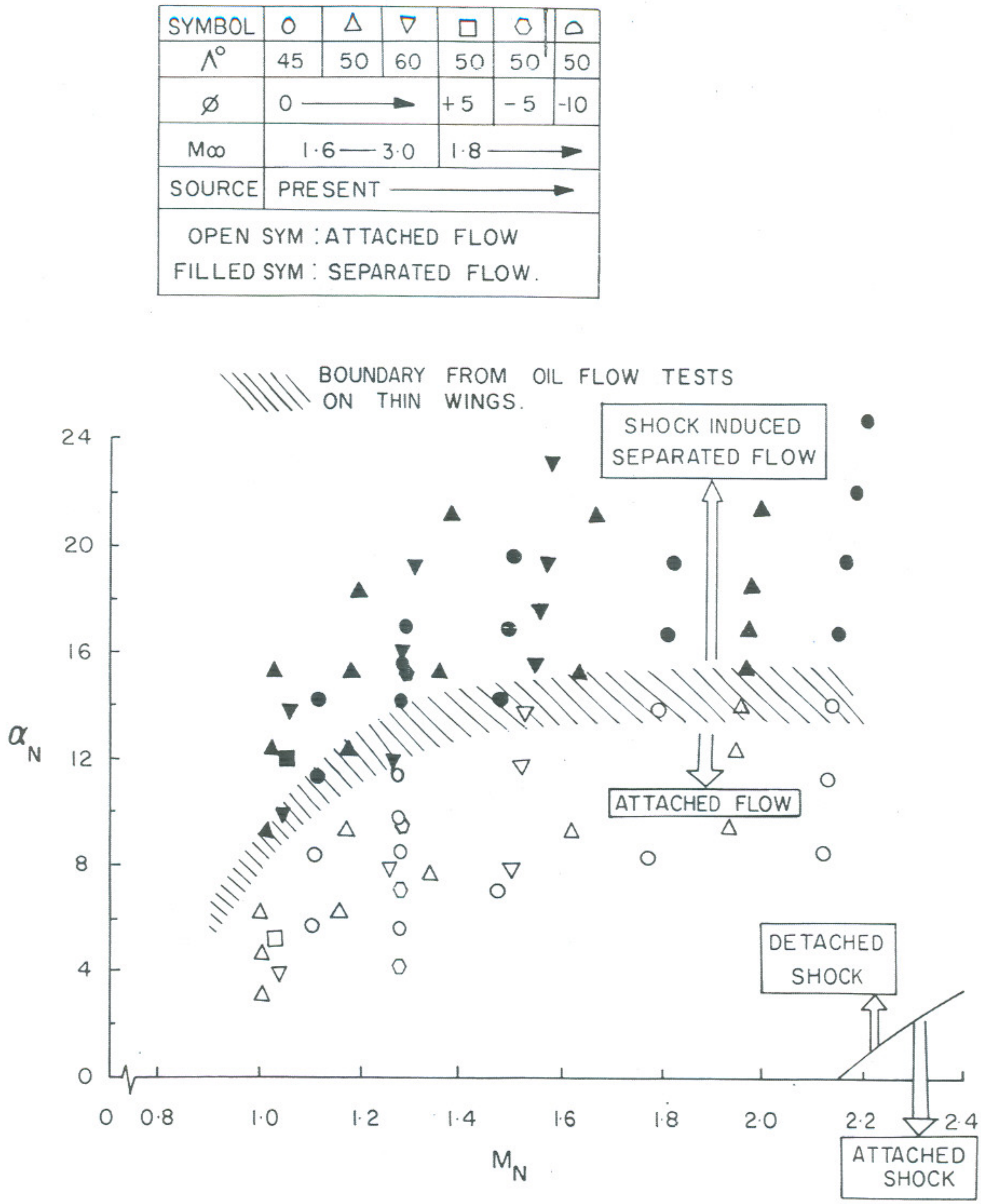


Figure 6. Experimental results from oil flow tests for thick wings (δ_N = 25°).

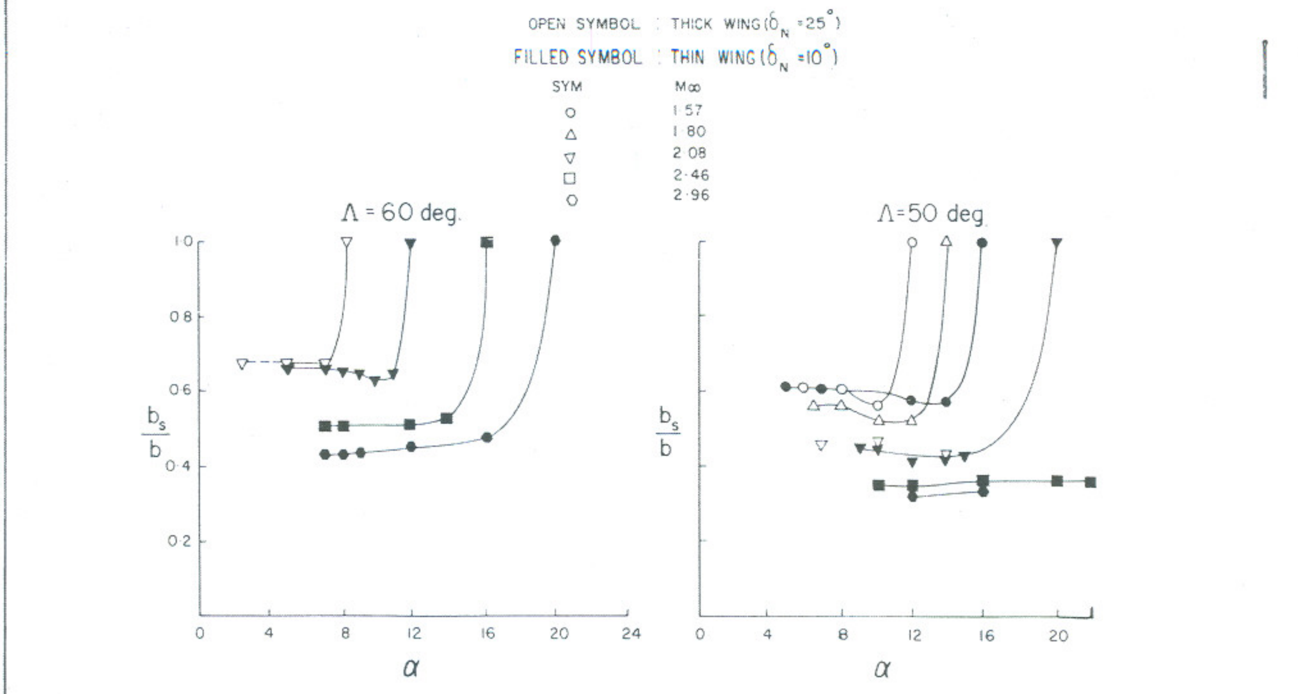


Figure 7. Separation line location.

Now, the attached flow at the leading edge is characterised by a swept Prandtl-Meyer expansion and since the condition upstream of the P-M fan is known, ie, M_{N1} (and thus υ_{N1}), the conditions downstream can be estimated since $\upsilon_{N2} = \upsilon_{N1} + \alpha_N$ (υ_{N1} is the Prandtl-Meyer angle corresponding to M_{N1}).

The Mach number and angle of attack normal to the leading edge are given by

$$M_N = M_\infty[1 - \sin^2\Lambda \cos^2\alpha]^{1/2} \quad . \quad . \quad . \quad (1)$$

and

$$\alpha_N = \tan^{-1}[\tan\alpha/\cos\Lambda] \quad . \quad . \quad . \quad (2)$$

The Mach number along the leading edge is given by

$$M_{T1} = M_\infty \cos\alpha \sin\Lambda \quad . \quad . \quad . \quad (3)$$

Across the P-M fan, the tangential velocity is conserved and thus,

$$M_{T2} = K\left[\frac{1+(\gamma-1)M_{N2}^2/2}{(1-(\gamma-1)K^2/2)}\right]^{1/2} \quad . \quad . \quad . \quad (4)$$

where M_{T2} = tangential Mach number downstream of the P-M fan, and

$$K = M_{T1}/(1 + 2M_\infty^2/(\gamma - 1)) \quad . \quad . \quad . \quad (5)$$

Knowing M_{N2} and M_{T2}, the resultant Mach number downstream of the P-M fan is

$$M_{R2} = (M_{N2}^2 + M_{T2}^2)^{1/2} \quad . \quad . \quad . \quad (6)$$

and the resultant flow direction relative to the freestream is

$$\theta = \tan^{-1}(M_{N2}/M_{T2}) - (\pi/2 - \Lambda) \quad . \quad . \quad . \quad (7)$$

The resultant flow at a Mach number of M_{R2} is turned by the embedded shock wave through an angle θ to align it to the chordwise direction. Our proposed criterion for incipient separation on the lee surface of a delta wing, is that

$$M_{R2}\theta = 0{\cdot}3 \text{ rad.}$$

(It may be noted that this is equivalent to a pressure rise across the embedded shock wave of 1·5.)

For a wing of sweep angle Λ at a freestream Mach number of M_∞, the above calculation is made at several angles of attack α, and an α_i determined where $M_{R2}\theta = 0{\cdot}3$ rad. At this value of α, incipient separation due to shock wave boundary layer interaction can be expected to occur. Figure 11 shows the results of this calculation made for wings with sweep angles of 45° to 65° and at Mach numbers from 1·6 to 3·0. The results in the form of a boundary between attached and inboard separated flows shows a small but unmistakable dependence on the sweep angle of the leading edge. At any M_N, α_N for incipient separation is higher for a wing with smaller sweep angle. Also plotted in this figure is the shaded band representing the experimentally determined boundary between these two types of flow from Fig. 5. It can be seen that there is a very good agreement between the two.

It was mentioned earlier that the boundary between the two types of flow was predicted under the assumption that the windward wide shock is attached. However, as seen in Figs. 5 and 6, all the experimental cases correspond to the case where the shock is detached from the leading edge on the windward side. The reason why the predicted boundary for attached shock case agrees well with experimental results for

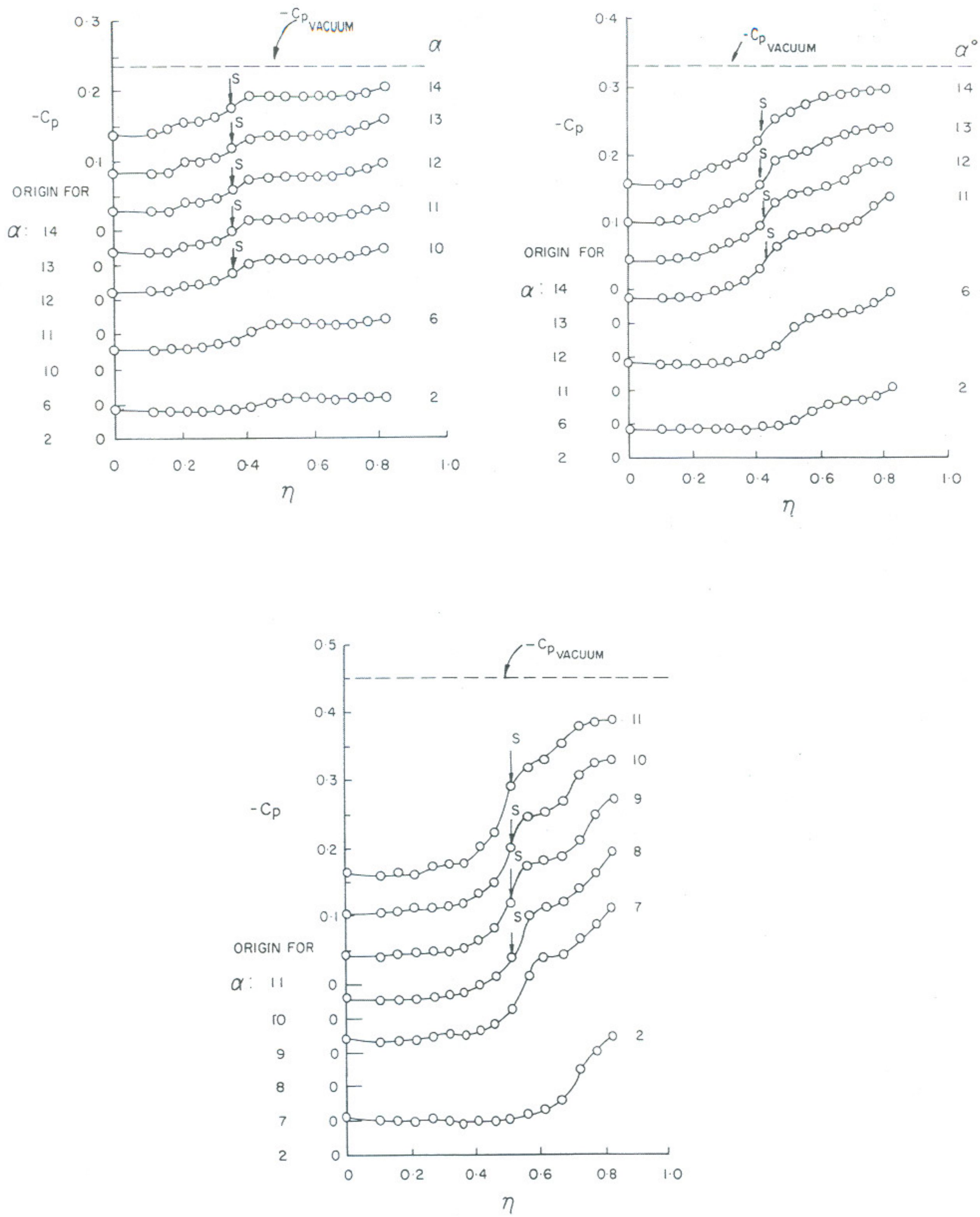


Figure 8. Lee surface static pressure distribution on 50° sweepback wing.
(a) M_∞ = 2·46 (b) M_∞ = 2·08 (c) M_∞ = 1·8.

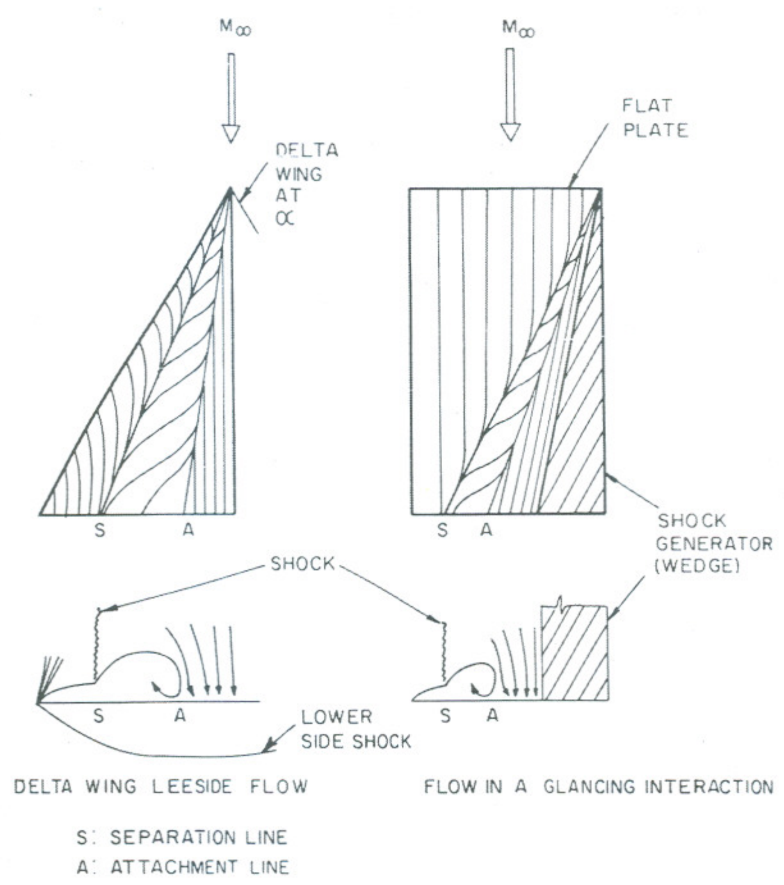


Figure 9. Similarity between delta wing leeside flow and flow in a glancing interaction.

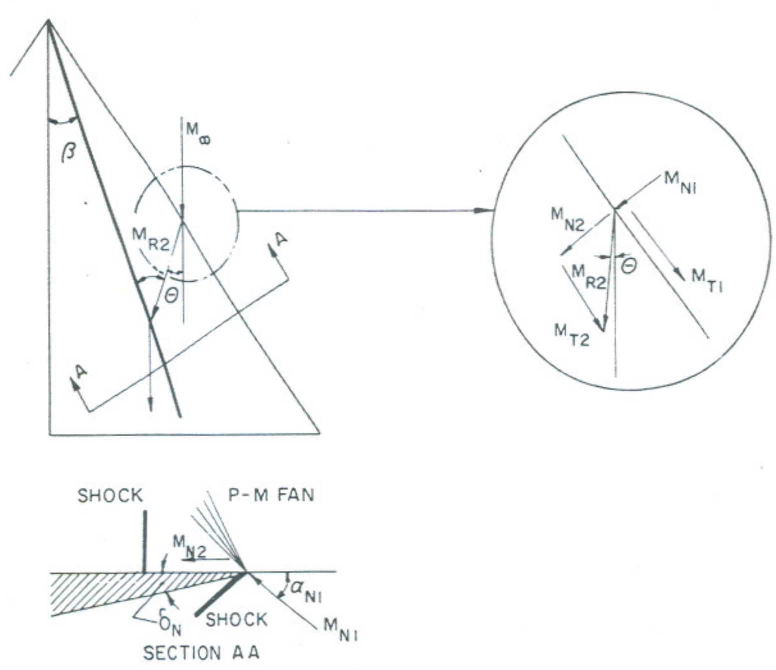


Figure 10. Flow geometry at the leading edge.

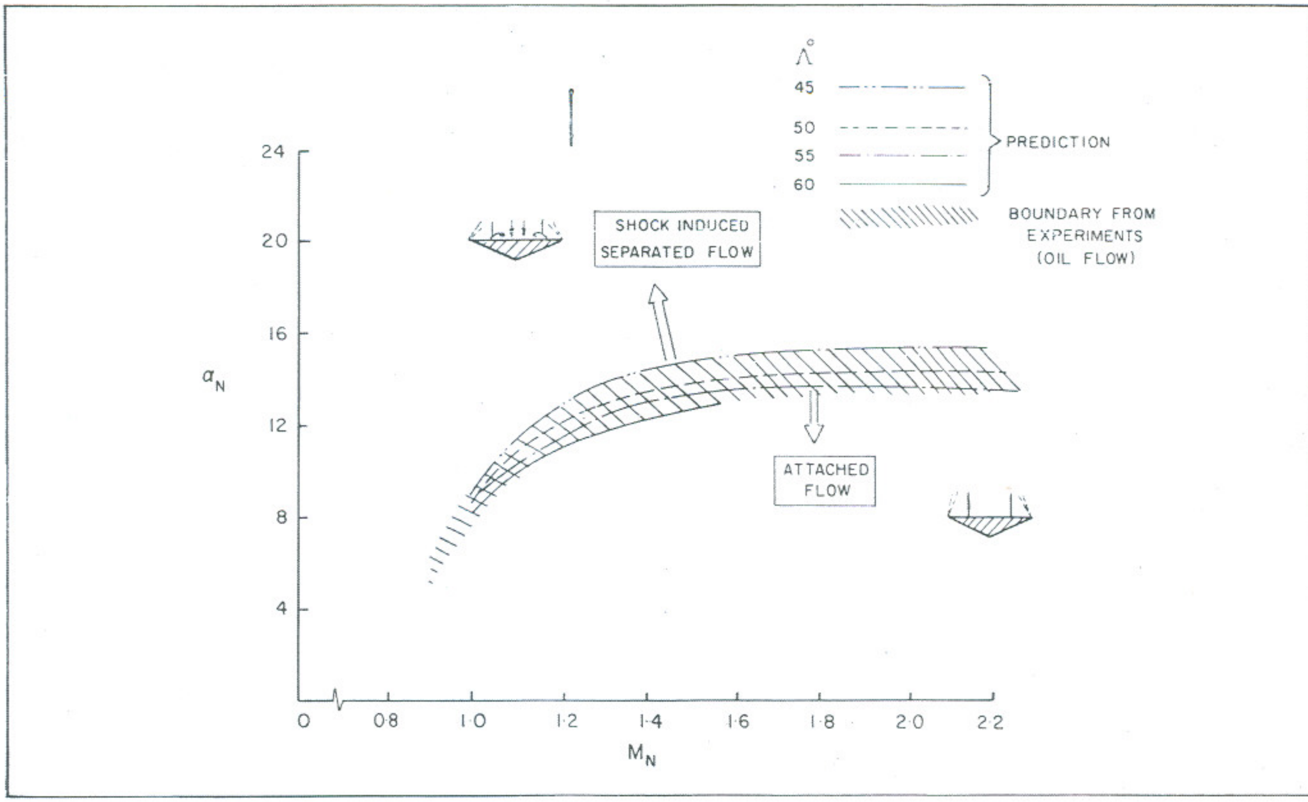


Figure 11. Comparison of predicted boundary with experimental results (oil flow tests) for thin wings.

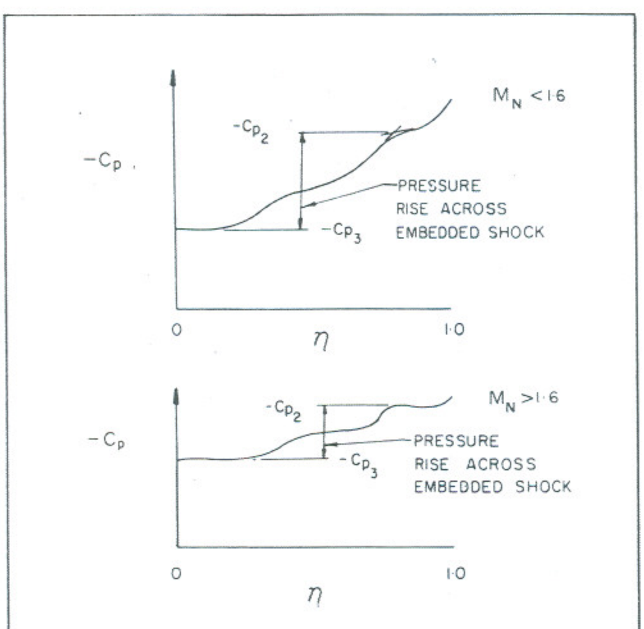


Figure 12. Definition of pressure rise across embedded shock wave.

detached shock cases could be the following. Squire[12] has compared experimentally obtained pressure distributions on the lee side of a delta wing supporting a detached shock wave with estimations made on the same wing under the assumption of attached leading edge shock. The results showed that although differences between measurement and estimation exists close to the leading edge, the estimated and the measured pressure distributions in the inboard regions agreed fairly well. This suggests that the state of the windward side shock may not be too critical to the flow in the inboard region on the lee surface. Thus, the boundary between the two types of flow predicted under the assumption of attached shock may be expected to be valid for cases with detached shock also.

In the tests carried out, Reynolds number varied from 4.5 millions to 7 millions (based on wing chord) depending on the Mach number. Assuming that the distance from the leading edge to the interaction zone (separation line location is a good measure of this) corresponds to approximately $\eta = 0{\cdot}5$ (*see* Fig. 7), the lowest Reynolds number based on this distance (to interaction) is about 1·0 million. This compares with the lower Reynolds number limit of about a million for the validity of the predicted boundary[11]. Due to facility limitations, no significant variation of Reynolds number was possible at fixed Mach numbers.

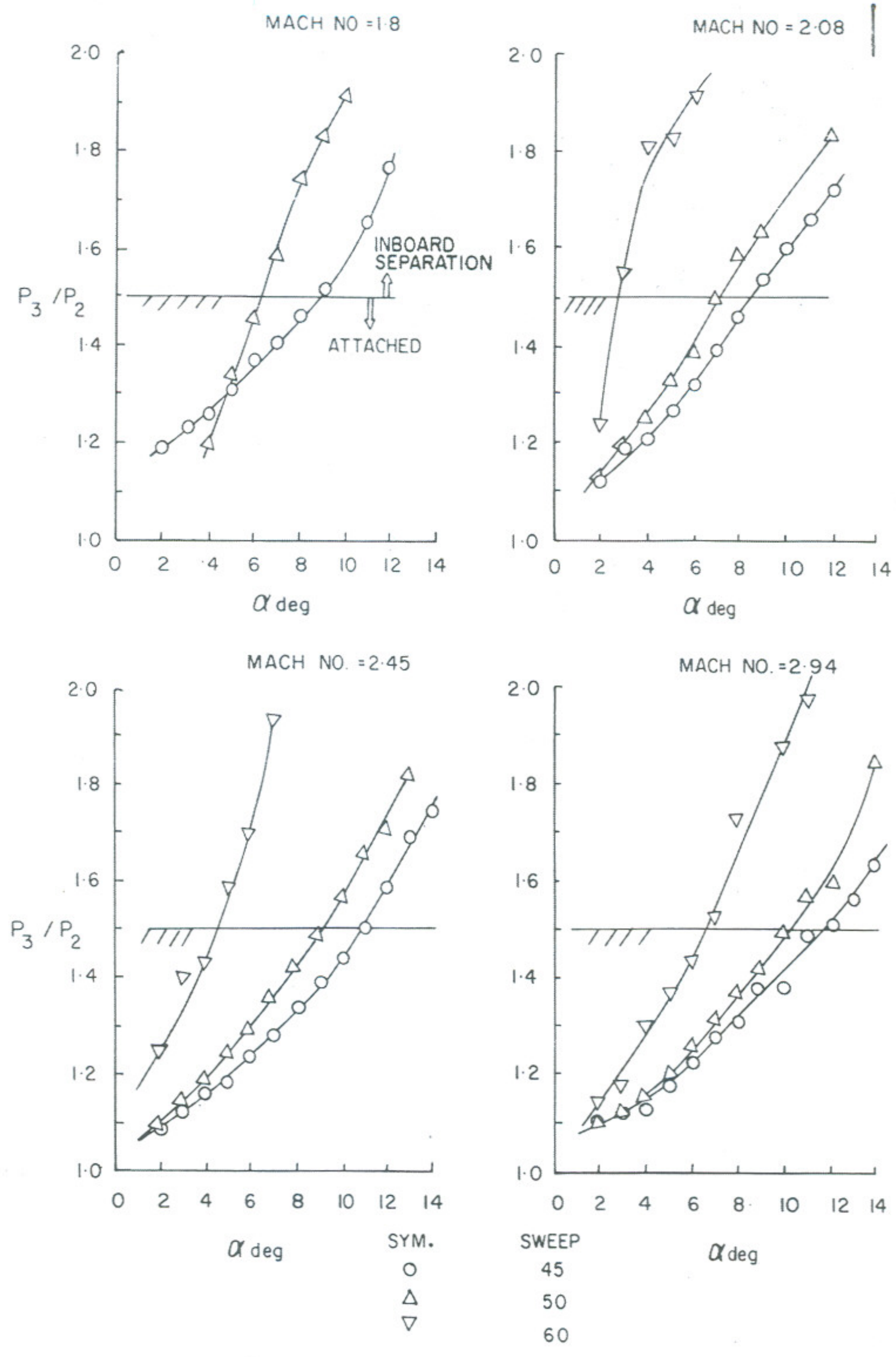


Figure 13. Variation of pressure rise across embedded shock with α, M∞, Λ.

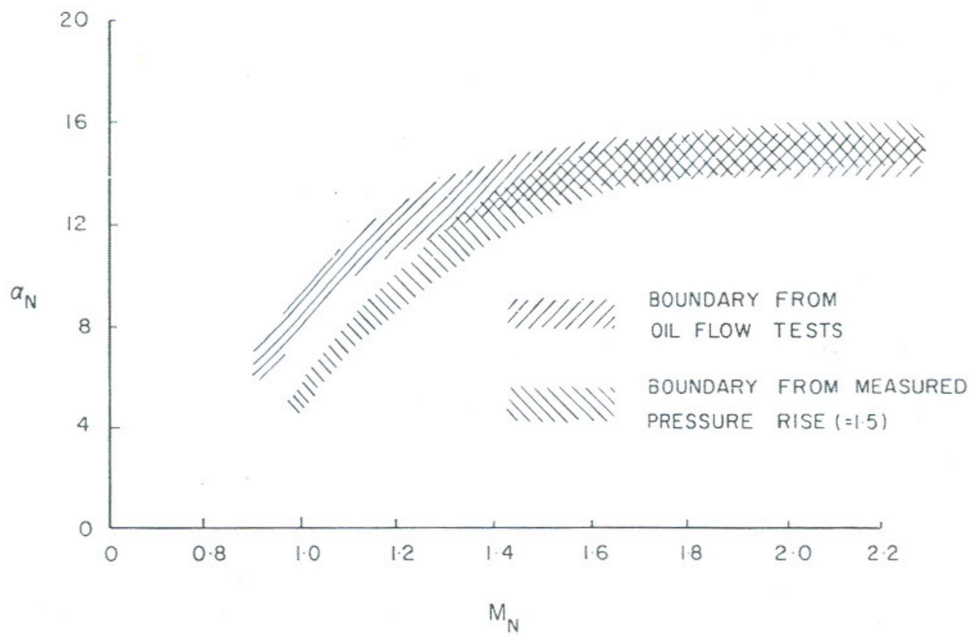


Figure 14. Comparison of boundary from oil flow tests with the boundary from measured pressure rise = 1·5.

5. CORRELATION OF SHOCK PRESSURE RISE AND SEPARATION

The pressure rise across the embedded shock wave can be estimated from the pressure distribution; the pressure rise is taken to be the ratio of the uniform pressures that exists near the centreline and outboard. The pressure rise associated with the reattachment of the leading edge bubble is not considered when estimating the pressure rise across the embedded shock wave. Figure 12 shows schematically the definition of pressure rise for the two cases: $M_N > 1{\cdot}6$ and $M_N < 1{\cdot}6$. Figure 13 shows pressure rise (p_3/p_2) plotted as a function of angle of attack for the various wings at different freestream Mach numbers. It is seen from this figure that for higher sweepback wings, the embedded shock is stronger, and its strength increases more rapidly with α at a given Mach number.

Now, having established that the predicted boundary using the criterion $M_{R2}\theta = 0{\cdot}3$ (or $p_3/p_2 = 1{\cdot}5$) agrees well with oil flow results, it would be interesting to see whether p_3/p_2 measured from the pressure distribution is in fact approximately equal to 1·5 at all points within the experimental boundary.

The results of this exercise are shown in Fig. 14, where the points at which $p_3/p_2 = 1{\cdot}5$ are plotted in a $M_N - \alpha_N$ plane. Also plotted in the same figure is the experimentally determined boundary (from oil flow patterns) between attached flows and flows with inboard separation (from Fig. 5). It is seen from the figure that the locus of points at which $p_3/p_2 = 1{\cdot}5$ obtained from the measured static pressure distribution lies within the experimentally obtained boundary between the two types of flow for $M_N > 1{\cdot}4$. The differences between the two for $M_N < 1{\cdot}4$ would seem to be due to the difficulty in measuring p_3/p_2 from the static pressure distribution, since it can be seen from Fig. 8(c), that there is no well defined uniform flow region inboard of the leading edge.

6. CONCLUSIONS

An experimental investigation has been carried out to study some features of shock-induced separated flows on the lee surface of delta wings. The boundary between fully attached flows and flows with shock-induced separation has been obtained from oil flow visualisations. The results indicate that this boundary is insensitive to wing thickness within the limit of thicknesses tested ($\delta_N = 25°$). It is shown that this boundary can be predicted by the criterion $M_{R2}\theta = 0{\cdot}3$ rad, derived from Korkegi's criterion for incipient separation in a glancing shock boundary layer interaction. The predicted boundary agrees well with the experimentally obtained one. Some features of the flow are discussed in terms of static pressure distributions and location of the separation line.

7. ACKNOWLEDGMENT

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